ENVIRONMENTAL ASSURANCE PROGRAM FOR THE PHOENIX MARS MISSION

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ABSTRACT

The Phoenix Mars mission involves delivering a stationary science lander on to the surface of Mars in the polar region within the latitude band 65°N to 72°N. Its primary objective is to perform in-situ and remote sensing investigations that will characterize the chemistry of the materials at the local surface, subsurface, and atmosphere. The Phoenix spacecraft was launched on August 4, 2007 and will arrive at Mars in May 2008. The lander includes a suite of seven (7) science instruments. This mission is baselined for up to 90 sols (Martian days) of digging, sampling, and analysis. Operating at the Mars polar region creates a challenging environment for the Phoenix landed subsystems and instruments with Mars surface temperature extremes between -120°C to 25°C and diurnal thermal cycling in excess of 145°C. Some engineering and science hardware inside the lander were qualification tested up to 80°C to account for self heating. Furthermore, many of the hardware for this mission were inherited from earlier missions: the lander from the Mars Surveyor Program 2001 (MSP'01) and instruments from the MSP'01 and the Mars Polar Lander. Ensuring all the hardware was properly qualified and flight acceptance tested to meet the environments for this mission required defining and implementing an environmental assurance program that included a detailed heritage review coupled with tailored flight acceptance testing. A heritage review process with defined acceptance success criteria was developed and is presented in this paper together with the lessons learned in its implementation. This paper also provides a detailed description of the environmental assurance program of the Phoenix Mars mission. This program includes assembly/subsystem and system level testing in the areas of dynamics, thermal, and electromagnetic compatibility, as well as venting/pressure, dust, radiation, and meteoroid analyses to meet the challenging environment of this mission.

KEYWORDS: Phoenix project, Mars mission, environmental assurance, environmental testing, extreme environments, thermal testing, dynamics testing, electromagnetic compatibility, environmental analysis, Mars lander

1. INTRODUCTION

The Phoenix Mars mission is the first of NASA's Mars Scout missions that was competitively selected in August 2003. Like the Phoenix bird of ancient mythology, the Phoenix mission is reborn out of fire - a new mission is created from the embers of previous missions, hence its project name. Phoenix returns to flight from the Mars Surveyor Program's 2001 (MSP'01) lander that was terminated.

The Phoenix mission has the primary objective of placing a science lander onto the Martian surface at a higher latitude than previous missions to perform in-situ and remote sensing investigations that will characterize the chemistry of the materials at the local surface, subsurface, and atmosphere and will identify potential provenance of key indicator elements of significance to the biological potential of Mars, including potential organics within any accessible water ice.

These objectives will be accomplished by landing in the north polar region (specifically between 65°N and 72°N), suspected to have a large reservoir of water, in the form of ice on the surface, and performing scientific analysis of the Martian arctic soils both in the surface and the near subsurface (<1m). Figure 1 shows a map of Mars indicating the landing region.

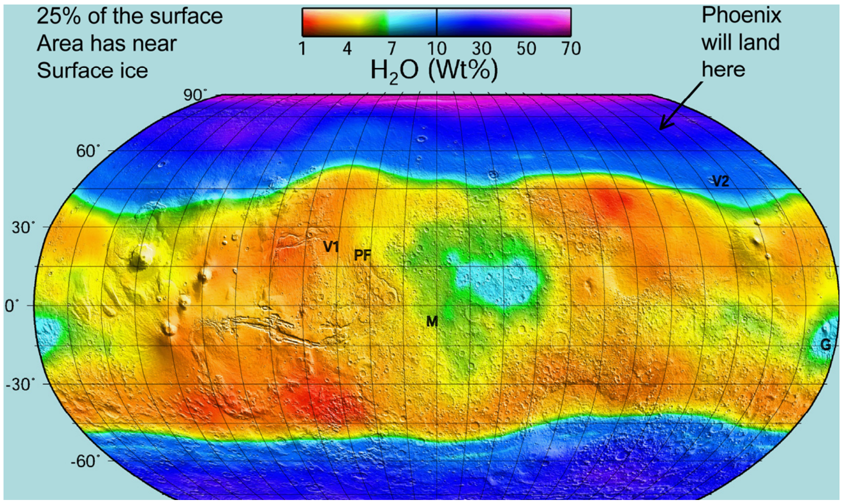

Figure 1. Mars Landing Region

The Phoenix spacecraft was launched in the early morning of August 4, 2007 atop a Delta-II rocket. The lander is scheduled to arrive on Mars on May 25, 2008 after a 9-month cruise. The landing time will be just prior to the start of northern summer Ls=76.7°, which optimizes the amount of solar energy available for the mission. The Phoenix baseline mission consists of up to 90 sols of digging, sampling, and analysis. The lander is expected to survive up to 150 sols on the surface when the combination of dust degradation on the solar panels and minimal available sunlight will not be enough to meet the power demands of the Lander and payload electronics, including the increased heating levels required to combat the plunging ambient temperatures.

1.1 Flight System Description

The Phoenix spacecraft is the MSP'01 Lander with all problem resolution "Return-to-Flight" recommendations incorporated. The Phoenix flight system consists of a cruise stage, heat shield, back shell, and lander, as shown in Figure 2. The heart of the flight system is the lander. With few exceptions, the lander is a fully redundant spacecraft with all subsystems necessary to deliver the Phoenix payload to the surface and support it through landed operations. The heat shield provides thermal protection during entry. The back shell provides thermal protection during entry, as well as support to the parachute. The back shell also provides mounting surfaces for Entry, Descent, and Landing (EDL) communications antennas. The cruise stage is essentially a launch adapter ring with only the necessary cruise hardware. The cruise hardware includes star trackers, solar arrays, telecom hardware, and sun sensors. All other spacecraft functions, including propulsion via thrusters threading through holes in the back shell, are performed by the lander.

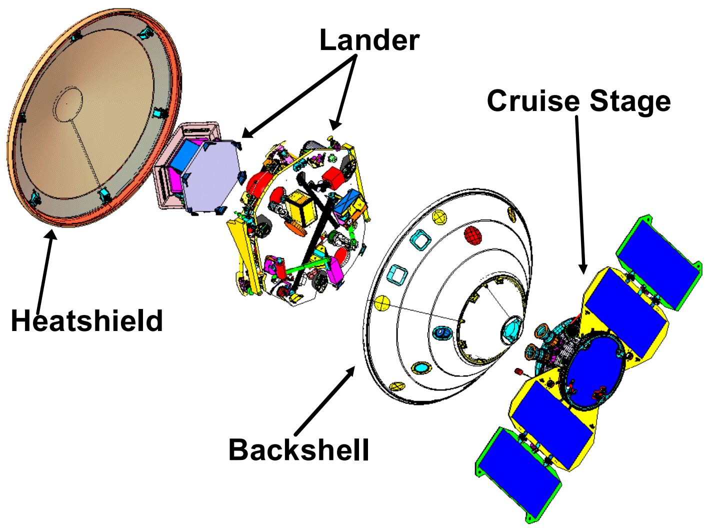

Figure 2. Phoenix Flight System

1.2 Payload System Description

The Phoenix science instruments/payload system consists of a suite of scientific instruments and tools for geologic exploration of the Martian surface. A list of the seven (7) instruments/payloads and their purposes are summarized below:

1. Mars Descent Imager (MARDI) - Provide landing site geological context of Martian surface during descent.
2. Surface Stereo Imager (SSI) - Characterize the landing and digging site; assess local geology to determine active processes that shape the terrain.
3. Robotic Arm Camera (RAC) – Take close up images of trench, dump pile, patch plate, scoop and rocks; choose & document samples and verify delivery.
4. Thermal and Evolved-Gas Analyzer (TEGA) – Consists of a Differential Scanning Calorimeter (DSC) and a Mass Spectrometer (MS). Determine the nature of volatile-bearing minerals and ices including the isotopic composition of gases evolved on heating. Determine the quantity of organic compounds in the soil or atmosphere (if any). Determine the molecular and isotopic composition of the atmosphere.
5. Microscopy, Electro-chemistry and Conductivity Analyzer (MECA) - (i) Wet Chemistry Laboratory (WCL): chemical analysis of soil; (ii) Microscopes: examine soil grains for evidence of liquid water interactions; (iii) Thermal & Electrical-Conductivity Probe (TECP): determine ice content, characterize thermal and vapor diffusion pathway, determine difficulty of melting ice; (iv) Patch Plate: study soil/atmosphere interactions.
6. Robotic Arm (RA) - Dig trench, position RAC and TECP (Thermal & Electrical-Conductivity Probe), acquire and deliver soil samples to TEGA & MECA, measure surface hardness.
7. Meteorological Station (MET) - Study local metrological conditions (pressure, temperature); detect clouds, dust plumes and dust devils; monitor boundary layer depth and dust profile.

The Phoenix payload elements are designed to meet the Phoenix science and measurement objectives. Figure 3 shows a conceptual depiction of the Lander during the digging of soil. The locations of the payload and spacecraft elements on the Lander deck are shown.

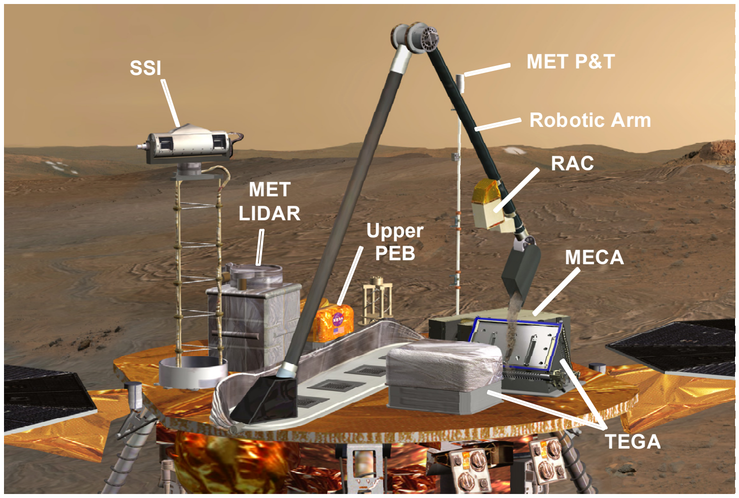

Fig. 3. Lander with Payload and Spacecraft Elements

2. ENVIRONMENTAL ASSURANCE PROGRAM

The objective of the environmental assurance program was to assure that Phoenix flight hardware was designed to survive and function during the extreme environments encountered in the ground operations, launch, cruise, Mars EDL, and Mars surface operations phases. It involved defining environmental requirements, supporting environmental testing and analyses, and verifying environmental compliance[1,2]. The environmental testing and analysis program was applied at both the assembly/subsystem and the system levels.

Environmental testing was the preferred method of environmental design verification. Analyses were performed for those mission environments that might be impractical to verify by test or that were more cost effective than testing, such as meteoroid compatibility, venting, and radiation dosage compatibility.

Flight hardware environmental testing verification was accomplished using three approaches: (1) qualification (Qual) test of an engineering model (EM) followed by flight acceptance (FA) testing of the flight models (FM) - used primarily on the payload instruments; (2) protoflight (PF) testing of all flight units, used primarily on payload instrument and selected inherited flight system hardware; and (3) a combination of PF and FA testing, used primarily on flight system hardware consisting of multiple units.

Qualification testing is normally performed on a dedicated Qualification Model or flight-like EM of the flight hardware, which is not intended to fly, in order to demonstrate the hardware design functions within specification for the maximum expected flight environments plus margins.

Protoflight (PF) testing is performed on flight hardware, which is intended to be flown, and for which there is no or inadequate previous qualification heritage. Protoflight testing accomplishes the combined purposes of design qualification and flight acceptance.

Flight Acceptance (FA) testing is typically performed on flight hardware and spares to verify flight workmanship quality when a previous design qualification test has been performed on an identical item.

3. ASSEMBLY ENVIRONMENTAL ASSURANCE PROGRAM

All Phoenix hardware was required to demonstrate compatibility and survivability in the dynamics, thermal, electromagnetic and natural space environments. Table 1 shows a summary of the environmental verification requirements for the assembly/subsystem and spacecraft system.

Table 1. Summary of Environmental Verification Requirements

Assembly/Subsystem	Spacecraft System
Dynamics tests • Random vibration (including frequency survey) • Pyroshock • Acoustic • Landing load	**Dynamics tests** • Low-level random survey • Quasi-static loads (launch/entry) • Acoustic noise • Pyro firing
Thermal tests • Thermal vacuum/atmosphere (all hardware) • Thermal Mars atmosphere (landed hardware) • Thermal cycling life qual (selected hardware)	**Thermal tests** • Thermal vacuum (w/ thermal balance - critical h/w at FA limits during functional)
EMC tests • Conducted susceptibility/emission • Radiated susceptibility/emission • Grounding & isolation • Multipacting/ionization breakdown (corona)	**EMC tests** • Radiated emission • Radiated susceptibility • Self compatibility • Magnetic cleanliness
Environmental analyses • Radiation (TID, DD, SEE) • Venting (pressurization & depressurization) • Meteoroid (performed at system level)	**Environmental analyses** • Orbital debris • Meteoroid (probability of survival & shielding) • ESD (touch down)

The Phoenix project relied heavily on heritage from MSP'01 and previously flown missions. The first step in the environmental verification of the Phoenix assemblies was to conduct an extensive review of the inherited requirements and hardware environmental test histories.

The Phoenix environmental requirements were inherited from the MSP'01 program. To ensure suitability of MSP'01 requirements for Phoenix, inherited thermal, dynamics, radiation, and EMC requirements were audited by the environments engineer to ensure their applicability to the environment to which Phoenix would be exposed during its mission to Mars.

The next step in developing the Phoenix environmental verification program was to obtain and review the inherited environmental test data for the heritage hardware and designs. The primary objective of this review was to ensure that the inherited thermal, random vibration, pyroshock, and electromagnetic compatibility as-tested levels either met or exceeded the Phoenix requirements with appropriate margins. Table 2 shows the environmental design and test margins used in testing and assessing the inherited environmental test data for Phoenix. Inherited hardware, which could not demonstrate sufficient test margins, required additional (delta) testing in order to be considered qualified for this mission. All inherited hardware had to demonstrate compatibility in both design and application to the Phoenix requirements. Depending on the extent of the deficiency, either a full protoflight or an abbreviated protoflight test was performed. Minor deficiencies in inherited environmental testing were waived or deferred to subsystem or system level verification. Table 3 shows a partial list of major Phoenix equipments, which were inherited from previous programs and the status of their environmental qualification.

Table 2: Environmental Design and Test Margin Requirements for Phoenix

<table>
<tr><th colspan="2">Environment</th><th>Design/ Qualification</th><th>Protoflight (PF)</th><th>Flight Acceptance (FA)</th></tr>
<tr><td colspan="2">Acoustics Level
Duration</td><td>MEFL + 3 dB
2 min</td><td>MEFL + 3 dB
1 min</td><td>MEFL
1 min</td></tr>
<tr><td colspan="2">Random Vibration
Level Duration</td><td>MEFL + 3 dB
2 min/axis</td><td>MEFL + 3 dB
1 min/axis</td><td>MEFL
1 min/axis</td></tr>
<tr><td colspan="2">Pyro Shock
Firings or Levels</td><td>MEFL + 3 dB
2 shocks/axis</td><td>MEFL + 3 dB
1 shock/axis</td><td>N/A
(no test required)</td></tr>
<tr><td colspan="2">Landing Loads</td><td>Design: apply Factors of Safety
Qual: MEFL x 1.2 (Test Factor)</td><td>MEFL x 1.2 (Test Factor)</td><td>MEFL</td></tr>
<tr><td rowspan="3">Thermal Vacuum</td><td>Temp.
Levels</td><td>Cold:
AFT - 15°C or -35°C
(whichever is colder)

Hot:
Non-electronics:
AFT + 20°C
Electronics:
AFT + 20°C or +75°C
(whichever is higher)</td><td>Cold:
AFT - 15°C or -35°C
(whichever is colder)

Hot:
Non-electronics: AFT + 20°C
Electronics:
AFT + 20°C or +75°C
(whichever is higher)</td><td>Cold:
AFT - 5°C

Hot:
Non-electronics: AFT + 5°C
Electronics:
AFT + 5°C</td></tr>
<tr><td>Test
Duration</td><td>Cold: 24 hours

Hot:
Non-electronics: 24 hours
Electronics: 144 hours</td><td>Cold: 24 hours

Hot:
Non-electronics: 24 hours
Electronics: 144 hours, 60 hrs. (subsequent units)</td><td>Cold: 24 hours
Non-electronics: 24 hours
Electronics: 8 hours

Hot:
Non-electronics: 24 hours
Electronics: 60 hours</td></tr>
<tr><td># of
Cycles</td><td>3 cycles
(cumulative)</td><td>3 cycles
(cumulative)</td><td>3 cycles
(cumulative)</td></tr>
<tr><td colspan="2">EMC</td><td>Min EFL - 6 dB
(emissions)
MEFL + 6 dB
(susceptibility)</td><td>Min EFL - 6 dB
(emissions)
MEFL + 6 dB
(susceptibility)</td><td>N/A
(grounding/
isolation only)</td></tr>
<tr><td colspan="2">Charged Particle
Radiation</td><td>TID, Displacement Damage: RDF = 2
Spot shielding,
RDF =3</td><td></td><td></td></tr>
</table>

Notes for Table 2:

MEFL = Maximum Expected Flight Level
Min EFL = Minimum Expected Flight Level
AFT = Allowable Flight Temperature
RDF = Radiation Design Factor

Table 3: Final Qualification Status of Major Inherited Equipment for Phoenix

Subsystem/Component	Design Heritage	Thermal	Random vibration	Pyro shock	Acoustic	EMC
Flight System						
C&DH Module	M01	Q_h/PF	PF	W	-	Q_h
Lithium Ion Battery	M01	Q_h/FA	Q_h/FA	ST	-	-
Lander Solar Array	M01	PF	PF	Q_h	ST	-
Cell Bypass Unit	M01	Q_h/PF	Q_h/PF	W	-	Q_h
Thermal Battery	MER	Q_h	Q_h	Q_h	-	-
Charge Control Unit	Odyssey	Q_h/PF	Q_h/PF	W	-	Q_h
Power Ditr. & Drive Unit	Odyssey	Q_h/PF	PF	W	-	Q_h
Pyro Initiation Unit	Odyssey	Q_h/PF	Q_h/PF	Q_h/PF	-	Q_h
UHF Transceiver/Diplexer	Odyssey	Q_h/FA	Q_h/FA	Q_h/W	-	Q_h/PF
UHF Helix Antenna	Odyssey	Q_h	Q_h	Q_h	-	-
SDST	Odyssey	Q_h/PF	Q_h/PF	Q_h	-	Q_h
SSPA	MER	Q_h	Q_h/PF	Q_h	-	Q_h/PF
MGA	MPL	Q_h/PF	Q_h	ST	-	-
LGA	MPL	Q_h/PF	Q_h/PF	ST	-	-
Helium Tank	MPL	Q_h	Q_h	-	ST	-
Pressure Regulator	MPL	Q_h	Q_h	Q_h	-	-
Pyro Isolation Valves	MPL	Q_h	Q_h	Q_h	-	-
Fuel Tank	MPL	Q_h	Q_h	-	ST	-
Cruise Solar Array Panels	MPL	Q_h/PF	-	-	ST	-
Star Tracker	MRO	Q_h/PF	Q_h/PF	Q_h/PF	-	PF
Sun Sensor Assembly	Odyssey	Q_h/PF	Q_h/PF	Q_h/PF	-	-
IMU	COY	Q_h/PF	Q_h/FA	Q_h	-	PF
Payload						
MECA Assembly	M01	Q_h/PF	Q_h/PF	PF	-	PF
MARDI	MPL, M01	Q_h	Q_h	Q_h	-	-

Q_h: Environment fully qualified by heritage
Q_h/PF: Partially Qualified by heritage/required additional (delta) PF testing
Q_h/FA: Fully qualified by heritage/New-build required FA testing
PF: Qualification by heritage not justified or heritage data not available, required PF testing
ST: Qualification deferred to next-higher level of integration or at system level testing
W: Testing waived
Q_h/W: Heritage qualification waived
-: Test not required

3.1 Assembly Dynamics Testing

Dynamics testing at the assembly level included random vibration and pyrotechnic shock. (Acoustics testing for the assemblies were deferred to the system level.)

3.1.1 Assembly Random Vibration Testing

Random vibration was performed on all hardware. Force limiting was allowed for random vibration testing and it was utilized primarily on payload electronic hardware. Powered-on vibration was required for all assemblies that were powered on during either the launch ascent or Mars descent phases. Major electronic boxes were powered on during random vibration testing. However, due to the short duration of the random vibration testing, it was difficult to monitor these assemblies for any anomalies or intermittent failures during vibration testing.

The Phoenix spacecraft was divided into eight (8) random vibration zones. The protoflight test requirements ranged from 9.6 g_{rms} to 31.6 g_{rms}.

3.1.2 Assembly Pyroshock Testing

Pyroshock testing was required for hardware exposed to pyrotechnic shock loading, whether the loading was self-generated or induced by external sources. The pyroshock sources were predominantly due to separation and deployment events, such as launch vehicle payload adaptor fairing separation, lander leg deployment, and backshell separation.

For inherited hardware, the shock environment was verified by reviewing the heritage shock qualification data. In cases of test deficiency, a protoflight–level shock test was performed on either a flight spare or the actual flight hardware. Pyroshock testing was waived (deferred to the spacecraft system level testing) if shock levels were below 500gs. All other hardware units without a valid qualification shock history were shock-tested as per the requirements in Table 2.

The Phoenix spacecraft was divided into ten (10) pyroshock zones depending on the shock levels. The protoflight test requirements ranged from 849g (Shock Response Spectrum, SRS) to 4800g. Assembly-level shock tests were performed using a shaker table.

3.2 Assembly Thermal Testing

The Phoenix assembly-level thermal test requirements varied depending on the type of assembly: electronics versus non-electronics (structures, mechanisms, optics). The main differences were that there were no maximum and minimum hot and cold temperatures for non-electronics; and that the hot test dwell time for non-electronics was significantly shorter. The PF and FA test temperature limits for each assembly were specified in detail in the Phoenix Temperature Table, which also contained all the allowable flight temperatures. Waivers were generated for all assemblies that did not meet the required test margins. All the landed assemblies (hardware that needed to operate on the surface of Mars) also required thermal testing in a low-pressure environment of CO_2 or GN_2 (6-10 Torr) to simulate the Mars surface pressure.

Tailoring of the thermal test requirements was frequently done for individual assemblies in order to optimize testing. The following list describes some deviations from the standard requirements:

a) Temperature atmosphere dynamometer testing was frequently substituted for thermal vacuum testing of actuators to improve performance characterization.
b) The verification focus for mechanical assemblies was on the fidelity of functional testing during environmental exposure rather than subjecting the mechanism to the standard 24/50 hours cold/hot durations typically applied to electronic hardware.
c) Non-operational PF test limits were augmented to encompass the planetary protection bake-out requirement (110°C or 125°C) for assemblies that were required to undergo Planetary Protection bake-out (dry heat microbial reduction) prior to system integration. This testing was important to assure that the assemblies would survive the high bake-out temperatures and be able to function properly afterwards.
d) Radio frequency (RF) assemblies were required to operate during GN_2 back-fill in order to demonstrate resistance to Corona break-down. For critical RF hardware, CO_2 testing was performed in a small bell jar at ambient temperature. Only the descent and lander antennas were subjected to Corona breakdown testing in CO_2 while at extreme temperatures.
e) Some tailoring was also allowed for the hot dwell qualification duration requirement (144 hours) for electronics. As the program got underway and the schedule became critical, some compromises were made in order to optimize functional testing and achieve a balance of hot and cold exposure.
f) Temperature atmosphere testing in lieu of thermal vacuum testing was authorized for some flight system assemblies that were not sensitive to vacuum. In these cases, hot temperatures were raised to compensate for the non-vacuum environment.

3.3 Assembly EMC Testing

The EMC testing of Phoenix hardware can be categorized into three phases. For flight system hardware, which were inherited from previous programs, an extensive review of the previous EMC/EMI testing was conducted to identify any potential deficiencies in the heritage test data. For minor deficiencies waivers were processed. For more severe deficiencies retests were recommended and performed at the assembly level.

In the case of the payload/instruments, an integrated payload EMC/EMI test was performed. The integrated test included the SSI, MECA, Robotic Arm, Robotic Arm electronics, Robotic Arm Camera, the TEGA, and the Metrology Station. For all the subsystems that were connected directly to the power bus, comprehensive EMI/EMC tests, including conducted and radiated emissions, conducted and radiated susceptibility, and isolation/grounding were performed.

The last category of EMC testing was performed at the system level for three spacecraft configurations, namely, (1) System Launch/Cruise EMI-EMC Radiated Self-Compatibility with the X-band, (2) self-compatibility of spacecraft bus, UHF link, and radar operation during terminal descent, and (3) self-compatibility of payloads with each other and with UHF communication on the surface.

3.4 Payload Assembly Environmental Testing

Payload engineering models or dedicated Qualification models were subjected to the entire suite of qualification tests. The payload flight units were subjected to thermal and random vibration tests. Table 4 shows a typical suite of assembly Qual/PF and FA tests for the Phoenix payload.

Due to the deep diurnal cycling of lander hardware on Mars, thermal cycling exposure was required for selected assemblies, generally performed at the electronic packaging level, although some tests were performed at the assembly level using engineering models.

Ideally, the tests to qualify the design should occur before the FA tests of flight hardware. However, because of the project's tight development schedule and the need to deliver the flight units for system integration, some of the environmental testing sequence was performed in a different order for a few pieces of hardware.

Table 4. Typical Suite of Assembly Environmental Tests for Phoenix Payload

	Qual or 1st PF unit:	FA or PF (subsequent units):
Random Vibration	✓	✓
Pyroshock	✓	---
Thermal vacuum	✓	✓
Thermal vacuum + Mars atmosphere (5 to 10 Torr GN_2)	✓	✓
EMC	✓	---

3.5 Assembly Environmental Analyses

The three main types of environmental compatibility analyses required at the assembly-level were pressure decay, Mars dust, and radiation. (Meteoroid survivability and shielding analyses were performed as system-level analyses.)

The verification to survive launch depressurization and Mars atmospheric entry repressurization was a simple venting analysis to ensure that vent holes were large enough to avoid trapped volumes.

Radiation analyses for total ionization dose (TID), displacement damage (DD), and single event effects (SEE) were performed for the MSP'01 project. The Phoenix radiation requirements were inherited from the MSP'01 project. An audit of the MSP'01 inherited radiation requirements showed that they were adequate for the Phoenix mission.

No formal electrostatic discharge analyses were required for Phoenix because the mission did not involve trapped radiation belts. The Phoenix lander uses hot gas rockets that will be operated continuously until touchdown. The hot gas of the rockets is ionized and will provide a slight, but adequate, conductive path that will equalize any voltage difference between the Phoenix lander and the surface of Mars. As a consequence, there will be no potential difference at landing and no touchdown ESD issues are expected.

3.6 Assembly Test and Analysis Metrics

Table 5 shows the number of different tests and analyses that were performed on a total of 106 Phoenix assemblies. These assemblies were testable units predominantly tested at the assembly level; a few of them, where appropriate, were also tested as multi-assemblies, essentially functional units consisting of several assemblies. In some cases, a single analysis was performed for more than one similar assembly. The number of tests shown in Table 5 includes testing at Qual, PF, and FA levels. Different dynamics tests were frequently performed at the same time using the same shaker table. The thermal tests shown in Table 5 include testing in vacuum, in atmospheric ambient, and in GN_2 or CO_2.

Table 5. The Number of Assembly/Subsystem Tests and Waivers

		Telecom	Propulsion	Command & Data Handling	Electrical Power Subsystem	Avionics	Payload	Thermal	Total
Assemblies		17	7	1	15	10	55	1 Note 1	106
Tests	Random Vibe	15	5	1	13	8	38	0	80
	Pyroshock	1	4	1	1	1	16	0	24
	Thermal	18	3	1	11	6	39	1	79
	EMC	0	0	0	0	0	1	0	1
Re-Tests	Dynamics	0	0	1	3	0	2	0	6
	Thermal	0	0	1	7	0	5	0	13
Venting Analysis		0	0	0	0	0	2	0	2
Waivers (Assembly)		11	3	5	16	4	17	5	61 Note 2
Waivers (System)		-	-	-	-	-	-	-	1 Note 3

Notes for Table 5:

(1) Except for the heat pipes, all other thermal control subsystem components were tested at system level.
(2) Breakdown of approved subsystem waivers is 13 dynamics, 38 EMC, and 10 thermal.
(3) Waiver to eliminate the system-level random vibration test.

Random vibration and thermal vacuum/atmosphere tests comprised the greatest number of tests. Venting analyses were the most common analyses. (The other analyses were inherited from other programs.) The retests were performed as a result of a previous failed test.

Environmental waivers were issued for various reasons. Of the 61 total environmental waivers issued for Phoenix, 38 of them were related to EMC issues. The latter group mainly dealt with deficiencies identified in the inherited EMC test data from MSP'01. Thirteen (13) environmental waivers were issued for dynamics-related deficiencies or test deferment to system level. The majority of deferred tests were pyroshock-related where expected flight shock levels did not warrant an assembly or subsystem level pyroshock test. Ten (10) waivers were issued for deficiencies in thermal requirements with one being a blanket waiver which reduced the thermal dwell duration for inherited assemblies from 144 hrs to 80 hrs.

One system-level environmental waiver was approved to eliminate the system-level random vibration test.

4. SYSTEM ENVIRONMENTAL PROGRAM

The objective of system-level environmental testing was to verify that the spacecraft in the launch, cruise, entry/descent, and landed configurations would perform within acceptable limits during and after exposure to the launch, cruise, EDL, and Mars surface environments.

Table 6 shows the overall system-level environmental test and analysis program that was implemented.

Table 6. System Environmental Program Summary

System-Level Test & Analysis Matrix for Phoenix Flight System	Environmental Verification Tests										
	Static Load (Mechanical)	Random Vibration	Acoustic	Pyro Shock	Modal Survey	Thermal Balance	Thermal Cycle (Vacuum)	EMI/EMC (Cond. Suscep.)	EMI/EMC (Cond. Emiss.)	EMI/EMC (Rad. Suscep.)	EMI/EMC (Rad. Emiss.)
System Level											
Launch Configuration	T	Waived	T	T	T	T	T	-	-	T	T
Cruise Configuration (Includes Backshell, Heatshield, Lander & Cruise Stage)	T	-	-	T	T	T	T	-	-	T	T
Entry and Descent Configuration	T	-	-	T	T	T	T	-	-	T	T
Landed Configuration	T	-	-	T	-	T	T	T	T	T	T

4.1 System Dynamics Testing

System acoustic testing was performed in the launch/cruise configuration (cruise stage, lander, heatshield, backshell). The purpose of dynamics tests was to validate the dynamics design in the launch, landing, and landed configurations to verify system functional tolerance to dynamics events, and to verify workmanship of the systems. Functional checkouts were performed before and after the environmental testing. All the test objectives were met.

In addition to acoustic testing, the spacecraft was exposed to critical deployment/shock events during cruise, backshell/heatshield separation, lander leg deployment, and MET Mast Release.

4.2 System Thermal Testing

System thermal testing was performed with hardware in two basic configurations: cruise and Mars surface operations. The objectives of these tests were: (i) to validate the thermal design in the cruise and landed configurations; (ii) to verify system functional performance at temperature at worst-case hot and cold extremes; and (iii) to verify workmanship of the systems. All the test objectives were met.

4.3 System EMC Testing

The purpose of system EMC testing was to confirm the self-compatibility of the integrated system and its compatibility with the launch, cruise, EDL, and surface operation electromagnetic environments. This testing was also used to qualify the pyrotechnic mechanisms and wiring for RF susceptibility reduction.

The Phoenix system EMC test program had the following specific objectives:

1. Demonstrate total functional performance for self-compatibility of the integrated system, including the interface cabling.
2. Demonstrate total functional performance of the integrated system for compatibility with the intended environment, including transmitter sources.
3. Demonstrate functional performance of the assemblies and their functions that were not tested in the assembly-level qualification test program.

5. LESSONS LEARNED

Numerous lessons have been learned from the implementation of the environmental assurance program for this mission that had a short project schedule. These include both technical and programmatic lessons. The following list includes the important overall lessons, which would be helpful in improving implementation of environmental programs for future projects.

1. The Phoenix project had a large quantity of hardware that was inherited from past missions. In some cases, the qualification and flight acceptance testing histories were not well understood. Significant effort by the team members was expended in determining which requirements had been met by earlier testing and which ones had not. If inherited hardware is to be used, a complete set of documentation should be gathered to provide evidence of compliance with the current project requirements.
2. Phoenix was a system-subcontract spacecraft. Several institutions were involved in building and testing of the Phoenix hardware. Some of these institutions have had a long history of building hardware for space flight with their own established environmental requirements processes and test philosophies. Finding a comfortable balance in implementing the Phoenix environmental requirements proved very challenging. There

must be an understanding that these contracting agencies intend to meet all the levied requirements with minimal oversight. This was necessary due to the tight environmental assurance budget for Phoenix.

3. A high-temperature bake-out, or Dry Heat Microbial Reduction (DHMR), at 110°C or 125°C, was required on most external lander hardware to reduce spore counts to meet NASA's planetary protection requirements. For Phoenix, the DHMR was typically the last step before delivering the flight hardware for integration with the spacecraft. Since the DHMR temperatures were higher than the nominal hot PF limit (+70°C) and there was no functional test after the bake-out, there was no certainty that the hardware would function after this process. Therefore, in cases where DHMR or similar high temperature exposure is required, it should be coordinated with the thermal vacuum test as a requirement. The bake-out or DHMR can be completed as the final cycle of the thermal vacuum test before the final functional checkout to ensure the hardware will function as required after the bake-out.
4. It was realized early in the project that, due to the limited environmental assurance oversight, it would not be possible to implement an environmental assurance program with consistent depth and penetration throughout the project. As a result, given the budget constraints and heavy heritage, it was decided to focus the main resources on verifying the pedigree of the inherited hardware and ensuring that a robust sub-system test program was implemented for Phoenix.

6. CONCLUSIONS

A comprehensive prelaunch environmental assurance program was successfully implemented on the Phoenix project. The rigorous environmental testing and analyses, coupled with a detailed inheritance review process, discovered a number of problems and anomalies, which have been resolved before launch.

7. CURRENT MISSION STATUS

Since its launch on August 4, 2007, the spacecraft has successfully completed two trajectory correction maneuvers (so far) that put it on course to be captured by the Martian gravitational field as it travels near the planet for landing. Additional trajectory corrections are scheduled for April and May of 2008 to further fine-tune the spacecraft's path, if necessary, to its landing site.

The Phoenix spacecraft has also made an orientation positioning adjustment to allow its solar panels to obtain more energy from the sun. During the first three months of the mission, the spacecraft's solar panels were not pointed directly at the sun to avoid the sun's power from overwhelming the spacecraft's electrical systems. The spacecraft's thruster engines were fired for a few seconds to point the solar panels more directly at the sun. This adjustment will continue as the spacecraft is moving further away from the sun towards Mars until the solar panels are working at full capacity.

All the subsystems have been functioning as expected with few deviations from predicted performance. There were limited numbers of flight operational anomalies. Key activities performed so far included checkouts of the radar and communication subsystems that would be used during and after the landing. Initial in-flight check-out of the mission's science instruments have been successful. Landing is scheduled for May 25, 2008.

8. ACKNOWLEDGEMENT

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BIOGRAPHIES

Dr. Kin F. Man is currently Supervisor of the Environmental Requirements Engineering Group in the Reliability Section at the Jet Propulsion Laboratory (JPL). He was the Environmental Requirements Engineer for the Mars Exploration Rovers Project. Dr. Man came to JPL in 1987 as a research scientist in the Atomic and Molecular Physics Group of the Science Division. He then went onto developing technologies for the microgravity program. For the past decade, he has been an environments lead for a number of NASA interplanetary missions, including the Prometheus and the Mars Science Laboratory Projects.

Alan Hoffman has forty-five years experience in analyzing and specifying environmental requirements at the Jet Propulsion Laboratory. He served as the Galileo (Jupiter Mission) Environmental Requirements Engineer for five years and the Cassini (Saturn Mission) Environmental Requirements Engineer for seven years. Since 1997 Mr. Hoffman is serving as advisor to flight projects during their development phase, mentoring environmental requirements engineers, and providing review board support. He also provides leadership in developing institutional standards and tools for environmental specification and verification.

Maher C. Natour has been at JPL since 2001, first as a mechanical reliability analyst for the Mars Exploration Rovers Project and then as the Environmental Requirements Engineer for the Phoenix and AMT Projects. Prior to JPL, Mr. Natour worked as a Mechanical Engineer for Space System/Loral in Palo Alto, California and the Nuclear Division of the Bechtel Corporation in San Francisco, California. Currently, he is providing environmental requirements engineering support to the Juno and GRAIL Projects.